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FLIGHT INVESTIGATION OF VTOL CONTROL AND DISPLAY CONCEPT FOR PERFORMING DECELERATING APPROACHES TO AN INSTRUMENT HOVER

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16. Abstract

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SUMMARY

It is generally acknowledged that the ability of VTOL aircraft to continue routine operation during periods of poor visibility is a necessary condition for economic feasibility in commercial usage. In the past, research efforts oriented toward VTOL terminal-area operation have attempted to attain adequate instrument flight capability by emphasizing either the display or handling qualities aspect of the problems, while paying relatively little attention to the other aspect. None of these studies, however, has demonstrated the capability of decelerating to an instrument hover at a preselected point.

This investigation resulted in a VTOL-oriented display and control concept which provided a capability of studying operational aspects of the decelerating approach problem. The flight tests were conducted with a research helicopter equipped with a control augmentation system and a flight-director display, along with more conventional situation displays. The tests documented the maximum deceleration rates that could be achieved, effects of winds, and approach and hover performance.

INTRODUCTION

It is generally acknowledged that the ability of VTOL aircraft to continue routine operation during periods of poor visibility is a necessary condition for economic feasibility in commercial usage. For such aircraft, landing operations will typically be conducted into confined areas to a landing pad without the benefit of a runway. The confined-area aspect of the problem imposes the need for precise guidance on steep inclined or, perhaps, even curved paths; the lack of a runway imposes a requirement for decelerating to hover, or near hover, along the flight path. Additional constraints, including efficient use of airspace in the terminal area and minimizing time spent at high power, dictate that the approach be made at a high speed, with the final deceleration being performed as rapidly as possible near the landing pad.

Past research has emphasized either the display or the handling qualities aspect of the VTOL instrument flight problem with relatively little attention being paid to the other aspect, the expectation generally being that sufficient capability could be achieved with one to compensate for deficiencies of the other. Bringing a vehicle to a hover at a preselected spot on instruments, however, has proven to be an extremely challenging problem, the ultimate solution for which will require a high degree of sophistication in both display and control technology. In particular, as discussed in reference 1, a variety of situation displays evaluated by the Langley Research Center has proven inadequate for performing instrument decelerating approaches to a hover. Whether used in conjunction with basic helicopter handling characteristics or with an advanced state-of-the-art rate-stabilization system, available situation displays resulted in an excessive scanning and mental workload for the pilot.

The purpose of the present investigation was to define a VTOL-oriented display and control concept which would permit the performance of inclined, decelerating approaches to an instrument hover. The scope of this investigation included implementation of the concept in a research aircraft and evaluation of the concept in terms of the effects of winds, maximum deceleration and desirable deceleration profiles, flight-path-tracking performance, and pilot workload. Since available situation displays have proven inadequate for this task, a flight-director system which displayed control-input commands to the pilot was implemented in a CH-46C research helicopter; and an attitude-command control system was also provided. Approaches were performed along a 6° flight path with a ground-based tracking radar system supplying position information to the aircraft via a telemetry link. The tests were initiated by performing instrument hover trials at the landing pad to define system requirements for the zero-speed end point. These tests were followed by decelerating approaches to a hover on instruments.

SYMBOLS

Values are given in both SI and U.S. Customary Units. The measurements and calculations were made in U.S. Customary Units.

A_1 pitching acceleration per unit pitch-control input, radians/second^2/centimeter (radians/second^2/inch)

A_2 pitching acceleration proportional to and opposing pitching velocity, stable when negative, second^{-1}

A_3 pitching acceleration proportional to and opposing pitch attitude changes, stable when negative, second^{-2}

a_Y — acceleration along body Y-axis, g units

B_1 — rolling acceleration per unit roll-control input, radians/second^2/centimeter (radians/second^2/inch)

B_2 — rolling acceleration proportional to and opposing rolling velocity, stable when negative, second^{-1}

B_3 — rolling acceleration proportional to and opposing roll attitude changes, stable when negative, second^{-2}

C_1 — yawing acceleration per unit pedal input, radians/second^2/centimeter (radians/second^2/inch)

C_2 — yawing acceleration per unit roll-control input, favorable yaw when positive, radians/second^2/centimeter (radians/second^2/inch)

C_3 — yawing acceleration proportional to and opposing yawing velocity, second^{-1}

C_4 — yawing acceleration proportional to and opposing unbalanced forces along the body Y-axis (unbalanced forces sensed by a body-mounted lateral accelerometer), stable when negative, $\frac{\text{radians/second}^2}{\text{meter/second}^2}$ $\left(\frac{\text{radians/second}^2}{\text{foot/second}^2}\right)$

C_5 — yawing acceleration proportional to and opposing heading changes, stable when negative, second^{-2}

$K_1 \ldots K_4$ — control system gains

p — roll angular velocity, radians/second

q — pitch angular velocity, radians/second

r — yaw angular velocity, radians/second

s — Laplacian operator

t — time, seconds

γ — flight-path angle, radians

δ_X roll control deflection, centimeters (inches)

δ_Y pitch control deflection, centimeters (inches)

δ_Z yaw control (i.e., pedal) deflection, centimeters (inches)

Θ angle between horizontal and aircraft roll axis measured in vertical plane (i.e., attitude), radians

ζ damping ratio

τ_1, τ_2, τ_3 control system time constants, seconds

Φ angle between line drawn in aircraft Y-Z plane parallel to horizontal plane and aircraft pitch axis (i.e., roll attitude), radians

Ψ angle between projection of aircraft roll axis on horizontal plane and reference line drawn in horizontal plane, (i.e., aircraft heading), radians

ω_n undamped natural frequency, radians/second

Subscripts:

h parameters with respect to test helicopter

m parameters with respect to model (i.e., mathematical model corresponding to desired aircraft characteristics)

ϵ difference between actual value and desired value of parameter (i.e., error)

The following relationships exist between certain symbols:

Pitch	Roll
$A_1 = K_1K_2$	$B_1 = K_1K_2$
$A_2 = \frac{1}{\tau_2}$	$B_2 = \frac{1}{\tau_2}$
$A_3 = K_2$	$B_3 = K_2$

A dot over a symbol indicates a derivative with respect to time.

DESCRIPTION OF EQUIPMENT

The major hardware, software, and data-instrumentation items employed in this investigation are described in this section. The hardware includes the test helicopter, the guidance system, and the cockpit display; the software includes logic for the control augmentation system, which provided the desired handling qualities, and the logic for the flight director commands. The signal flow diagram in figure 1 indicates the interfaces of the major subsystems that were employed in this investigation. A ground-based tracking radar system sensed aircraft position. This position information was continuously relayed to the analog computer onboard the aircraft via a telemetry link. The onboard computer processed these signals for driving the guidance displays, which included a flight director, a radar altimeter, moving map, and an ILS indicator. The computer also received electrical inputs from the evaluation-pilot's controls, processed them in accordance with the programed control characteristics, and drove the helicopter control surfaces through electrohydraulic actuators.

Test Helicopter

The CH-46C research helicopter shown in figure 2 served as the test vehicle. The conventional mechanical control system on the right side of the cockpit was modified to "fly-by-wire" to facilitate the control implementation. The conventional control levers – consisting of a center stick, pedals, and a collective stick for height control – were retained. During simulated instrument flight, the evaluation pilot's view of the outside world was prohibited by masking of selected glass areas and use of an instrument hood, which attached to his helmet. The safety pilot's controls, located on the left side, were unaltered, and he could overpower or disengage the evaluation pilot's inputs at any time. The helicopter's physical characteristics are given in reference 2.

Guidance System

Aircraft position was sensed by a precision-tracking radar system located at the NASA Wallops Station, where the tests were performed. The position information was transmitted to the aircraft via a telemetry link in terms of rectangular coordinates, with the origin fixed at the landing pad and the X-axis alined with the localizer center line. Computation of aircraft position relative to a 6° flight path was performed by the onboard computer. A detailed description of the tracking-radar system is given in reference 3.

Cockpit Displays

A photograph of the research instrument panel is shown in figure 3. This panel included an attitude-director indicator (ADI), a radar altimeter, an instantaneous vertical-speed indicator, a moving-map display, and an ILS-type indicator. The power-lever position indicator, used only for establishing proper initial conditions for the tests, was not used during the approaches. Pertinent characteristics of the nonstandard displays are discussed in the following sections.

Attitude-director indicator.- The flight-director control commands were displayed on an attitude-director indicator (ADI), a close-up photograph of which is shown in figure 4. The ADI also displayed aircraft attitude in pitch and roll. As indicated by the figure, separate commands were provided for pitch, roll, and height control. By following these commands – that is, by moving the appropriate control in such a manner as to keep the needles centered – the pilot should be able to keep the aircraft on the prescribed path and at the prescribed speed throughout the final approach, which terminated in an instrument hover at the pad. For example, pitch-control inputs that were required to center the horizontal needle kept the aircraft at or returned it to the desired ground speed. Similarly, motions of the roll control and height control which would keep the vertical needle and side-mounted tab centered, respectively, resulted in the aircraft remaining on or returning to the desired flight path. The control characteristics that were provided for the aircraft yaw degree of freedom negated the need for presenting flight-director pedal commands. The flight-director logic is presented in appendix A.

Moving-map display.- A photograph of the moving-map display is shown in figure 5. The map moved vertically to display x-range to the pad and laterally to indicate lateral displacement with respect to the desired ground track, which defined the X-axis for the system. Since the map itself was not free to rotate, aircraft heading relative to the localizer course, or ground track, was represented by the cursor, which was indicated as an extension of the aircraft symbol at the center of the display. A dual scale factor was used such that, as range was reduced through 760 m (2500 ft), the map scale factor changed from 120 m/cm (1000 ft/in.) to 12 m/cm (100 ft/in.).

Radar altimeter.- The radar altimeter displayed absolute altitude relative to the landing-pad elevation. A nonlinear scale was used to provide a sensitive indication below 30 m (100 ft) while still maintaining an altitude range of 370 m (1200 ft).

Control Augmentation System

Experience has shown that conventional stability augmentation systems at low speed, although providing satisfactory stability in the classical handling-qualities sense, generally lack sufficient gain for adequately coping with external disturbances, vehicle trim changes and coupling, and, the varying characteristics of the vehicle during the performance of dynamic tasks. A control augmentation system is a type of control system which can offer a means for mitigating these effects; in such a system the aircraft response to pilot inputs is determined by a set of model dynamics. The manner in which this system was implemented is presented in appendix B, along with a presentation of the system frequency response characteristics for both pilot inputs and external disturbances.

By using the control augmentation system concept, an attitude-command control system was implemented for the pitch and roll degrees of freedom, and pilot-selectable heading-hold and automatic turn-following modes were implemented for the yaw degree of freedom. The helicopter response characteristics for the three translational degrees of freedom were unmodified. The control response characteristics for the four independently controlled degrees of freedom (pitch, roll, yaw, and vertical) are presented in table I; the characteristics tabulated for the vertical-translational degree of freedom were estimated from flight data.

Data Instrumentation

During the research flights, data were recorded at the radar ground station and onboard the aircraft. The ground-base data consisted of plots of altitude against range, cross range against range, and range rate (i.e., x-component of ground speed) against range. Onboard the aircraft, control position, flight-path deviations, range-rate error, position of the flight-director command needles, and aircraft attitudes were recorded on magnetic tape. Angular rates, accelerations, and other standard measurements were recorded on an onboard oscillograph.

RESULTS AND DISCUSSION

Instrument Hovering

The hovering trials were initiated by transferring control from the safety pilot to the hooded evaluation pilot with the aircraft established in a hover at an altitude of about 15 m (50 ft) and near the center of the pad. Centering the horizontal and vertical needles of the flight director would result in the aircraft remaining at or, if displaced, returning to the center of the pad; similarly, centering the side-mounted tab would keep the aircraft at or returning to the preselected hover altitude. Two minutes of records were obtained during each trial.

In the interest of simplifying the logic for the flight-director display, the decision was made to attempt the use of a fixed set of flight-director gains for the entire decelerating approach task. Since the zero-speed case was believed to be the critical condition, the instrument hovering trials were performed first. The flight-director gains thus established are shown in table II.

Figure 6 shows the hovering performance during two 2-min hovering trials. The one on the left, labeled IFR, was performed by reference to the instruments; the one labeled VFR was performed by visual reference (the view partially restricted by the masked glass areas) with the pilot looking outside and is offered for comparison purposes. From a number of such instrument trials the conclusion was that a point on the aircraft could be maintained consistently within a circle 11 m (35 ft) in diameter throughout the run. By way of comparison, the largest dimension of the aircraft is 24 m (80 ft).

The fact that the pilot could hover on instruments for adequately long periods indicates that the physical workload was at a reasonable level. Furthermore, the motions of the aircraft seemed steady and well controlled to the engineer-observer team onboard the aircraft. For example, the maximum angular rates encountered were only about 4°/sec and the average translational drift rate was less than 1 knot. However, the pilot did indicate the need for more confidence in the commands.

During other hovering tests, a qualitative assessment of the pilots reliance on the flight-director information was obtained while hovering by reference to the situation displays alone (i.e., the command needles were disabled). During these hovering tests over the pad, the pilot quickly lost position control, and the runs were typically aborted at a range of 61 m (200 ft) from the pad, with the aircraft still diverging. It was concluded, therefore, that flight-director information was necessary for hovering.

Decelerating Approaches

Task description.- The decelerating approaches were initiated by control being transferred to the evaluation pilot with the aircraft at an altitude of about 180 m (600 ft), on a heading of 30° to 45° relative to the localizer, and at an airspeed of 50 to 55 knots. The localizer was acquired at a range of about 2400 m (8000 ft) by reference to the moving map initially, then by centering the vertical needle (roll command) on the flight director as the needle began to come off the peg. At this point, the pilot began to center the horizontal needle (pitch command) which resulted in a ground speed of 45 knots. The pilot held constant altitude until the tab (height-control command) began to move off the upper stop, indicating that intercept with the 6° glide path was imminent as the aircraft approached a range of 1800 m (6000 ft). From this point on, the pilot continued the approach by attempting to keep the flight-director needles centered while simultaneously monitoring the situation displays. The velocity profile commanded the pilot to fly a ground speed of 45 knots until, at the appropriate range, the velocity profile corresponded to a constant deceleration. Near the pad, the velocity profile departed from a constant deceleration and transitioned into a linear relationship between command velocity and range. The hover altitude was offset 15 m (50 ft) above the pad during these tests. The run was terminated in a hover when the evaluation pilot was satisfied that the hover condition was reasonably well stabilized.

Deceleration limits encountered.- Starting with zero deceleration rate (i.e., constant speed approaches, which are reported separately in ref. 4), the approaches were performed with progressively higher deceleration rates to determine maximum rates and identify limiting factors. A total of over 100 decelerating approaches were performed, of which 80 percent were satisfactorily completed to a hover over the pad. It should be emphasized, however, that the 20 unsuccessful approaches represented the identification of limitations which were being sought. As a result of this exploration, it would now be possible to specify conditions for which one could predict a very high probability of a successful approach. Aborts would represent "go-around" conditions.

Examination of the unsuccessful decelerating approaches revealed a common cause which, in every case, was related to pilot reluctance to follow the flight-director commands when doing so would produce large pitch attitudes. For the helicopter, where speed is controlled through attitude changes, the entire vehicle must be pitched up to decelerate. In general, whenever the flight director commanded control inputs which resulted in a nose-up attitude greater than about 12°, the pilot tended to ignore the commands, partially because of a lack of confidence in the system. Ignoring the speed commands for any period of time caused the error to increase still further. Although he might eventually get the needle centered as the aircraft neared the pad, the distraction caused by this needle being displaced would usually result in poor tracking of the altitude command. The resulting deviation from the glide path caused the approach to be aborted on a couple of occasions.

In a hover, the aircraft attitude is about 8° nose-up. With the pilot limiting his control commands to 12° nose-up, there was only a 4° margin available for decelerating. This 4° margin above the hover attitude corresponds to a horizontal deceleration slightly greater than 0.08g ($1g = 9.8$ m/sec^2) when one includes the contribution from profile drag. In fact, initial attempts at flying a velocity profile corresponding to the 0.08g deceleration were generally unsuccessful because there was no attitude margin for correcting deviations about the nominal profile. However, the lead information provided by the addition of a normal-accelerometer input to the flight-director logic, discussed in the next section, improved pilot confidence to the extent that 0.08g decelerations became acceptable. During visual tests, maximum decelerations of approximately 0.15g were achieved.

It should be emphasized that the deceleration limits encountered cannot be generalized for application to other VTOL aircraft which do not decelerate by changing pitch attitude. Also helicopters exhibiting a different absolute relationship between pitch attitude and deceleration would be expected to exhibit a different deceleration limit.

Tracking performance.- Figures 7(a), 7(b), 7(c), and 7(d) show composite plots of the performance obtained during the final 760 m (2500 ft) of 22 decelerating approaches that were obtained with the system at its most developed condition. These four figures represent approaches flown by the two research project pilots at two levels of longitudinal deceleration (0.06g and 0.08g). The deceleration level, the pilot by code, and the relative prevailing surface wind is given in each figure. Each figure is divided into three parts, with the upper part being a plot of ground speed against range; the middle, lateral deviation against range; and the lower, altitude against range. The dashed lines correspond to the desired level of the parameters as a function of range. In generating these performance plots, the traces were terminated at the point where the velocity reached zero. The subsequent motions that occurred in the vicinity of the pad are indicated by the brackets which show the maximum and root-mean-square excursions in both position and ground speed.

Perhaps worth mentioning for the sake of indicating the type of problems that may be encountered in a dynamic situation is a glide-path dropout problem which was experienced during earlier tests while the flight-director logic was still being developed. The dropout is clearly evident in figure 8. It was determined that the dropout was caused by the aircraft power-required characteristics, which necessitated a large and rapid power increase as the aircraft decelerated. The dropout occurred despite a 2-sec washout time constant on the power-lever (height-control lever) input to the flight-director logic and an attempt to provide lead information based on a simplified approximation of the aircraft power-required characteristics. Eventually it was found that sufficient lead could be provided to eliminate the dropout by adding a normal-accelerometer input to the flight-director logic. The accelerometer signal replaced both the power-lever input and the power-required lead used initially. This modification became a permanent part of the flight-director logic, which is shown in appendix A. The effect of this modification in curing the dropout can be seen by comparing the altitude-range plots of figure 8 with those of figure 7, which were obtained with the improved logic.

However, the replacement of the pilot-input term with the accelerometer term resulted in increased pilot effort in nulling the flight-director tab. Whereas for the pilot-input term the tab responded as if it were connected directly to the height-control lever, thereby making control of the tab very easy; for the accelerometer term the tab would not center, of course, until the aircraft had developed a normal acceleration. Thus, if the pilot conscientiously attempted to null the tab, there was a tendency toward overcontrolling. It is believed that this overcontrol tendency might be cured by the inclusion of control-rate information to the flight-director logic.

Figure 9 shows time histories of control activity, flight-director commands, aircraft motion, range-rate error, and flight-path deviation corresponding to one of the decelerating approaches presented in figure 8. (Unfortunately the corresponding time histories for the approaches shown in fig. 7 were lost because of a malfunction of the onboard magnetic tape recorder). The time histories are arranged to show, in order, for each controlled degree of freedom, the error in the variable being controlled, the control command displayed to the pilot, the pilot's response to the control command, and the aircraft response to the control input.

Time histories such as those of figure 9 are useful in analyzing the sequence of control activity, noise on the control commands, overall system performance, and the relative magnitude of the control task with respect to the various degrees of freedom. The figure indicates, for example, that very little pedal activity existed during the approach, as was anticipated for the heading-hold mode. It also indicates that, of the other controlled degrees of freedom, the vertical degree of freedom apparently required the least attention, or at least it received the lowest priority until very near the pad.

Handling Characteristics

Pitch, roll, and yaw degrees of freedom.- The high-gain command-control system employed during this investigation contributed substantially to the success achieved in performing decelerating approaches to a complete hover. In addition to providing precise angular response to pilot-control inputs, the high-gain system effectively decoupled the controlled degrees of freedom and virtually eliminated angular response to external disturbances. As a result of this high level of stabilization, pilot control activity about the angular degrees of freedom was required primarily for the relatively long-term guidance problem as opposed to control for guidance plus high-frequency stabilization of attitude. Reference to figure 9, for example, indicates that the amplitude of the high-frequency control activity for both pitch and roll is always less than 0.5 cm (0.2 in.). Even this small amplitude control motion probably does not represent needed or intentional control, since aircraft motions tend to feedback into the control stick through inertia of the pilot's arm. Also during performance of precision tasks, where the pilot's internal gains are operating at a high level, the pilot's sampling of the response contributes to the total control activity.

Heading hold versus turn following.- The two control options provided in yaw required some exploration to determine the best point in the approach for switching from turn following to heading hold. It was already known from the hovering trials that the heading-hold mode resulted in minimum pilot workload and, in fact, was required for hovering. On the other hand, it was known from a vast amount of flight experience that automatic turn following would be desirable while intercepting and acquiring the localizer. During these tests, therefore, the turn-following mode was used during acquisition of the flight path, but switching to the heading-hold mode was left entirely to pilot discretion and experimentation.

During the initial approach trials, the pilot adopted a technique whereby he would use the turn-following mode during most of the approach, switching to the heading-hold mode just prior to the range where the deceleration would begin. It was gradually realized, however, that localizer tracking improved and that pilot effort in the lateral degree of freedom simultaneously decreased when he switched to the heading-hold mode. He subsequently modified his technique whereby the heading-hold mode was selected shortly after capturing the localizer. In other words, it was found that at the low approach speeds used, the heading-hold mode permitted the pilot to readily make small lateral corrections by sideslipping the aircraft without disturbing the heading. One of the pilots, who has a broad base of experience in display research for low-speed instrument flight, commented that the localizer tracking performance and the associated level of pilot effort required with this system were superior to that experienced with any system he had flown.

Vertical degree of freedom.- No stability augmentation was provided for the vertical-translational degree of freedom, but the basic aircraft has a reasonable amount of normal velocity damping. Therefore, a control input to this degree of freedom resulted in a steady-state vertical rate. Although the vertical rate response characteristic of the aircraft was believed to be satisfactory in itself, considerable benefit would be realized from use of an augmentation system for this degree of freedom capable of compensating for the aircraft power-required characteristics.

Effects of Wind

Although all the decelerating approaches presented in figure 7 were made in the presence of cross winds, approaches were also performed with both head winds and tail winds. Of these conditions, the head winds, as might be expected, were the most desirable, and all the head-wind approaches were completed to a hover. The effects produced by the other wind conditions merit further discussion.

Tail winds.- Tail winds created a problem analogous to that encountered with higher deceleration profiles. In the presence of tail winds, the hover attitude increases about 0.1° per knot of wind, thereby directly reducing the attitude increment which the pilot is willing to use during the decelerations. It appeared that a 10-knot tail wind would reduce the maximum usable deceleration from 0.08g to about 0.06g.

Cross winds.- For the sake of simplicity, the flight-director logic used during this investigation assumed small angle differences between aircraft heading and localizer course. It was necessary, therefore, to compensate for high cross winds by banking the aircraft rather than by excessive crabbing. Although small crab angles (up to 15°) were acceptable from a system standpoint, extremely large crab angles (up to 90°) are needed to compensate for a given cross wind as the aircraft speed is reduced to zero.

As a consequence of using bank angle to compensate for cross winds, the maximum cross wind relative to the aircraft heading which could be tolerated was limited by the maximum steady bank angle that the pilot was willing to use. This bank angle appeared to be about 4^o to 5^o. For such steady-state bank angles, the pilot was aware of the side force and also became concerned about having to modulate the attitude about a mean bank angle of 5^o.

Operational Considerations

The most important result of this investigation has been definition and demonstration of a control and display concept which enabled the pilot to perform decelerating approaches to an instrument hover at a preselected spot. In particular, the merit of providing flight-director commands and their potential usefulness in VTOL operations were clearly evident. Also, the role of the control augmentation system in minimizing the pilots physical workload was evidenced by his comments regarding the absence of trim changes and coupling and was substantiated by analysis of the control activity and aircraft angular motions. Despite these accomplishments, however, limitations in pilot acceptance of the system were disclosed by pilot commentary in relating these tests to an operational environment. It appeared that pilot acceptance of the system was adversely influenced by poor display integration, sensor noise, and linear constraint on the variables being tracked. Each of these factors is discussed as follows.

Need for integrated display.- Although the attitude-director indicator was itself an integrated display in that it presented both attitude-situation information and flight-director commands, the pilot required additional situation information for validating the flight-director commands. This additional situation information (including ILS information, radar altitude, airspeed, and the moving-map display) was presented, but the physical separation of the information on a number of displays rendered the cross-check capability inadequate. The increasing use of cathode-ray tubes in advanced situation displays (ref. 5, for example) offers the potential for properly integrating situation and command information. Techniques exist with these displays for electronically generating control-command symbols and superimposing them on the situation information.

Effect of sensor noise.- The principal source of noise on the flight-director display came from the tracking radar system. Including filtering provided by the response characteristics of the flight-director instrument itself, the velocity input to the flight director was filtered by an amount equivalent to a first-order filter having a time constant of about 1.2 seconds. Even with this much filtering, the random needle motions were considered objectionable and contributed significantly to the pilot reluctance to track the needles tightly. In essence, the pilot was forced to act as a final filter. Mistrust of the validity of the signals was probably an important factor in his limiting the pitch attitude which, in turn, limited the maximum deceleration rate. It would appear desirable to keep the effects of sensor noise to less than 2 percent of full scale on the flight director.

Effect of linear constraint on tracked parameters.- In these tests a linear relationship existed between error in the parameter being tracked (e.g., glide path, localizer, or range rate) and the input to the flight director; in other words, twice as much error would double the needle deflection. Although this might appear reasonable, it tends to overwork the pilot by causing him to fly in a manner which is more precise than he would do under visual conditions. To illustrate this point, for example, during a deceleration under visual conditions, the pilot does not constrain himself to follow a velocity profile that is precisely defined at each point in space. Rather, he initiates a deceleration which he modulates to the extent necessary to achieve the final constraint (zero speed at zero range). Although computer techniques exist which would permit this type of control under instrument conditions, the flight-director logic could become extremely complex. It is thought that a similar easing of pilot workload might be simply achieved by requiring only that the error be maintained within some small range. Thus, if the desired speed is 45 knots he might be directed to make a disproportionately small control input when the actual speed is between, say, 43 to 47 knots. A similar criterion should probably exist for control and tracking of glide path and localizer so that he would be commanded to fly within a narrow corridor rather than along a line.

In order to achieve as high a degree of precision as possible near the pad, the sensitivity of the flight-director needles must be quite high. As a result, the needles are extremely active throughout the entire approach; thus, the pilot is directed to make numerous small control inputs. The high needle activity also gives the pilot the impression that the approach is sloppy, whereas in reality, it may be more precise than necessary. It would appear desirable, therefore, to reduce gains used for the approach to perhaps one-half the level used for hover.

CONCLUSIONS

Tests were conducted with a research helicopter to assess the capability provided by a VTOL control/display concept for performing decelerating approaches to a hover under instrument conditions. On the basis of these tests the following conclusions are drawn:

1. Flight-director information was essential for performing approaches to a hover at the pad.

2. The attitude-command stabilization system provided adequately satisfactory handling characteristics for the angular degrees of freedom, but additional stabilization for the vertical-translational degree of freedom would be desirable to improve approach precision and reduce pilot workload and apprehension.

3. The heading-hold mode provided superior handling qualities and capability for performing the final approach as compared with the turn-following mode, although the latter mode was still required during acquisition of the flight path.

4. Horizontal deceleration rates were limited by pilot reluctance to use large pitch attitudes; rates up to 0.08g could be performed in the presence of zero to light winds. (This is approximately one-half the maximum deceleration rate achieved during visual operations.) Different deceleration limits would be expected for aircraft which do not decelerate by changing pitch attitude.

5. Pilot reluctance to use large pitch attitudes reduced the maximum usable deceleration rate in the presence of tail winds. Cross-wind components relative to the aircraft heading greater than 10 knots were unacceptable because side forces acting on the pilot caused disorientation and apprehension.

6. In order to provide an operational capability, further research is required to refine the flight-director logic and to improve integration between the flight director and situation information.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., December 14, 1970.

APPENDIX A

FLIGHT-DIRECTOR LOGIC

In formulating the flight-director logic, the intent was to provide control command information, which, if followed by the pilot, would cause the aircraft to follow a prescribed flight path at a prescribed velocity. The outer loop parameters to be controlled, therefore, were altitude with respect to the desired altitude profile, which was specified as a function of range to the hover point; lateral deviation with respect to the desired ground track; and velocity, or range rate, with respect to a desired velocity profile, which was specified as a function of range to the hover point. The logic was somewhat simplified by the fact that only the low-speed range (-15 to 60 knots) was considered. The response to control was assumed sufficiently uncoupled, therefore, to permit the use of pure height-control inputs to control altitude and the use of pure pitch-control inputs to control speed. Similarly pure roll-control inputs were used to control the ground track using either of the yaw control modes. The flight-director logic required for operation at higher speeds has been investigated for a helicopter in reference 6.

Block diagrams illustrating the flight-director logic are shown in figures 10, 11, and 12 for the pitch-, roll-, and height-control commands, respectively. As indicated in figure 10, the pitch-control command was generated by taking the difference between the pitch-control-position error signal and a high passed control-position signal. The pitch-control-position error was formed by summing a signal proportional to range-rate error with a high passed nominal pitch-attitude signal. The range-rate-error signal represented the difference between the desired range rate, produced by the velocity profile function generator, and the actual range rate, which was derived in the onboard analog computer by using a standard approximate differentiation network. The nominal pitch-attitude signal was provided by a function generator which computed the nominal pitch attitude (i.e., the control-position trim point) required to stay on the programed velocity profile. High pass filtering of the nominal pitch-attitude signal was incorporated to remove steady-state components which would otherwise mask the long-term range-rate errors.

High pass filtering of the control-position signal was also incorporated to prevent a steady-state pitch-control-position signal from masking the pitch-control-position error signal.

The logic for the roll-control command (illustrated in fig. 11) also employed an approximate differentiation network to compute the lateral-deviation rate and high pass filtering of the roll-control-position signal to eliminate steady-state control-position inputs.

APPENDIX A

The height-control-command logic illustrated in figure 12 was similar to the roll-control-command logic except that the deviation signal was formed onboard the aircraft and a body-mounted normal-accelerometer signal was used instead of the height-control-position signal. (The significance of using the accelerometer signal as opposed to the control-position signal is discussed in the text.) High pass filtering of the normal-accelerometer signal is required since the accelerometer will sense a component of gravity which varies as a function of the trim attitude.

APPENDIX B

DESCRIPTION OF CONTROL AUGMENTATION SYSTEM

Background

Control augmentation systems have recently appeared in several military aircraft including the F-111, the A-7, and the C-5. Conceptually, the control augmentation system represents an extension of the model-following simulation technique which has been used successfully in flight research (ref. 2) for many years. The advantage of the control augmentation system over more conventional stabilization concepts is its capability to replace the aircraft basic response characteristics with an idealized set of characteristics. The idealized characteristics, referred to as the model, are selected on the basis of the handling qualities desired for performing a given mission. These characteristics are formulated as a set of model equations, which are programed on an airborne computer. The computer is electrically signaled by pilot control inputs, and the model response (i.e., the desired response) is continuously computed in real time. Standard control techniques, involving the use of lead and feedback compensation, are employed to force the aircraft to follow the desired response.

Description of Mathematical Models

This section describes the models used for the pitch, roll, and yaw degrees of freedom.

Pitch model.- The mathematical representation of the desired pitch response was

$$\dot{q}_m = A_1\delta_Y + A_2q_m + A_3\Theta_m \tag{1}$$

where

$$\Theta_m = \int_0^t\left(q_m \cos \Phi_m - r_m \sin \Phi_m\right)dt \tag{2}$$

Equation (1) defines a second-order system wherein a control step-input results in an attitude change. In the actual inplementation of the system, it was assumed that $\Theta_m \approx \Theta_h$ so that the model became

$$\dot{q}_m = A_1\delta_Y + A_2q_m + A_3\Theta_h \tag{3}$$

Although the assumption that $\Theta_m \approx \Theta_h$ results in some loss in mathematical precision, it greatly simplified the mechanization of the model. Specifically it avoids the need for computing model attitude (eq. (2)) and substitutes in its place a direct measurement of aircraft attitude from conventional instrumentation. The loss of mathematical precision is reflected by the fact that the usual method for computing damping ratio for a classical second-order system will not yield perfect results. Analytical examination of these phenomena has indicated that loss of mathematical precision is a function of the gains achieved in the compensation, with the actual damping ratio being somewhat less than the computed value. In the present investigation, however, the discrepancy was negligible.

Roll model.- The form of the roll model was identical to that of the pitch model, with the same assumptions and limitations discussed for pitch being applicable. The mathematical representation of the roll model was

$$\dot{p}_m = B_1\delta_X + B_2p_m + B_3\Phi_h \tag{4}$$

Yaw model.- The yaw model provided two modes (turn following and heading hold) represented mathematically as follows:

Turn following

$$\dot{r}_m = C_1\delta_Z + C_2\delta_X + C_3r_m + C_4a_{Y,h} \tag{5}$$

Heading hold

$$\dot{r}_m = C_1\delta_Z + C_3r_m + C_5\Psi_h \tag{6}$$

In the turn-following mode, side force, which is indicative of sideslip, was sensed by a body-mounted lateral accelerometer in the aircraft and used to provide the last term in equation (5). However, since adequate gain could not be achieved on this term to provide adequate turn-following performance because of limit cycle and noise problems, the yaw model was further coupled with lateral degree of freedom by the term containing the coefficient C_2, which provided favorable yaw. As indicated by equation (6), the heading input to the model for the heading-hold mode was obtained from a direct measurement of aircraft heading.

Implementation of Control Augmentation System

Having selected the desired model characteristics and formulated them in a suitable manner, the next step is to develop the compensation that will force the aircraft to follow the model. In this study, the compensation used included lead, rate error, and integrated rate error. Reference to figure 13, which is a block diagram of the entire system including the test helicopter, will aid in understanding the previous discussion of the model and the following discussion of the compensation. Although figure 13 applies to pitch and roll specifically, implementation of the yaw degree of freedom was identical with respect to the compensation, and the differences in the model implementation are directly deducible by comparing equations (5) and (6) with equation (3).

Lead compensations.- The lead compensation was based on the model angular acceleration and knowledge of the test aircraft response to control-surface inputs determined during calibration of the test aircraft. In effect the amount of control-surface motion required to produce a given angular acceleration was known to a fair approximation. The use of this lead compensation only would be roughly equivalent to a conventional stabilization system.

Rate-error compensation.- The rate-error compensation was provided by computing the error between the model angular rate and the aircraft angular rate. This error signal was fed into the aircraft control system on the highest possible gain, which was limited by noise and/or limit cycle, depending on the degree of freedom. If adequate gain could be achieved on this error signal no other form of compensation would be required to get the aircraft to follow the model accurately. Increasing the gain increases the aircraft frequency bandwidth, which means that the aircraft will accurately reproduce command motions up to that frequency, within the acceleration limits of the aircraft. The accepted rule of thumb is that the aircraft frequency bandwidth should be three to five times the model frequency bandwidth, which was 2 rad/sec for the pitch and roll models. Based solely on the rate-error gains attained, the aircraft frequency bandwidths were 4.5 and 3.5 rad/sec for pitch and roll; thus, there was a factor of only about two between the model and aircraft frequency bandwidth.

Integrated-rate-error compensation.- The integrated-rate-error compensation was obtained by integrating the rate error. The gain of this signal was also adjusted to the maximum permitted by limit cycle. This compensation provides the extremely high gain at low frequencies that suppresses the effect of slowly varying disturbances such as aircraft trim changes and disturbances due to wind shears.

Frequency Response

Figures 14 and 15 indicate the frequency response characteristics of the control augmentation system for the pitch and roll degrees of freedom, respectively. In each figure is plotted three response curves: (1) model attitude response to pilot input, (2) aircraft attitude response to a model attitude command, and (3) aircraft attitude response to an external disturbance acting on the fuselage and having an amplitude corresponding to 1.0 rad/sec^2. These frequency response plots were generated analytically, based on the transfer functions and gains shown in figure 13.

As indicated by figures 14 and 15, the model frequency bandwidth was about 2 rad/sec for both pitch and roll, and the high frequency roll-off was approximately that of a second-order system (40 dB/decade). By way of comparison, the aircraft bandwidth, after the addition of lead, was about 90 rad/sec and 10 rad/sec for pitch and roll, respectively. It is apparent, therefore, that the aircraft should accurately reproduce the frequency content of any signal which the model was capable of commanding. With respect to the aircraft response to external disturbance, it is apparent from figures 14 and 15 that very low frequency disturbances, such as trim changes and wind shears, would not disturb the aircraft attitude.

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3. Gracey, William; Sommer, Robert W.; and Tibbs, Don F.: Evaluation of Cross-Pointer-Type Instrument Display in Landing Approaches With a Helicopter. NASA TN D-3677, 1966.
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5. Dukes, Theodor A.: An Integrated Display Concept for Helicopters and VTOL Aircraft. No. 314, 25th Annual National Forum Proceedings, Amer. Helicopter Soc., Inc., May 1969.
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TABLE I.- CONTROL SYSTEM CHARACTERISTICS

Pitch and roll:	
Control power	2 rad/sec^2
Control sensitivity	$0.24 \frac{rad/sec^2}{cm}$ $\left(0.6 \frac{rad/sec^2}{in.}\right)$
Damping ratio	0.75
Undamped natural frequency	2.0 rad/sec
Attitude sensitivity	0.06 rad/cm (0.15 rad/in.)
Yaw:	
Heading-hold mode:	
Undamped natural frequency	2.0 rad/sec
Damping ratio	0.7
Maximum heading rate capability*	0.8 rad/sec
Heading rate control sensitivity*	$0.14 \frac{rad/sec}{cm}$ $\left(0.35 \frac{rad/sec}{in.}\right)$
Turn-following mode:	
Control power	0.25 rad/sec^2
Control sensitivity	$0.08 \frac{rad/sec^2}{cm}$ $\left(0.2 \frac{rad/sec^2}{in.}\right)$
Directional stability	$0.01 \frac{rad/sec^2}{m/sec}$ $\left(0.004 \frac{rad/sec^2}{ft/sec}\right)$
Damping-to-inertia ratio	0.7 sec^{-1}
Yaw due to lateral control	$0.026 \frac{rad/sec^2}{cm}$ $\left(0.065 \frac{rad/sec^2}{in.}\right)$
Vertical (approximate):	
Maximum thrust-to-weight ratio	>1.1
Height-control sensitivity	0.08 g/cm (0.2 g/in.)
Normal-velocity damping-to-mass ratio	0.4 sec^{-1}

*Outside a ±0.64-cm (±0.25 in.) deadband.

TABLE II.- FLIGHT-DIRECTOR GAINS

	Full-scale values
Horizontal needle:	
Range-rate error	±7.6 m/sec (±25 ft/sec)
Position error	±76 m (±250 ft)
Longitudinal-stick position	±5 cm (±2 in.)
Vertical needle:	
Rate of closure with respect to localizer	±7.6 m/sec (±25 ft/sec)
Localizer error	±76 m (±250 ft)
Lateral-stick position	±5 cm (±2 in.)
Vertical tab:	
Rate of closure with respect to glide path	±4.6 m/sec (±15 ft/sec)
Glide-path error	±46 m (±150 ft)
Normal acceleration	±0.23g

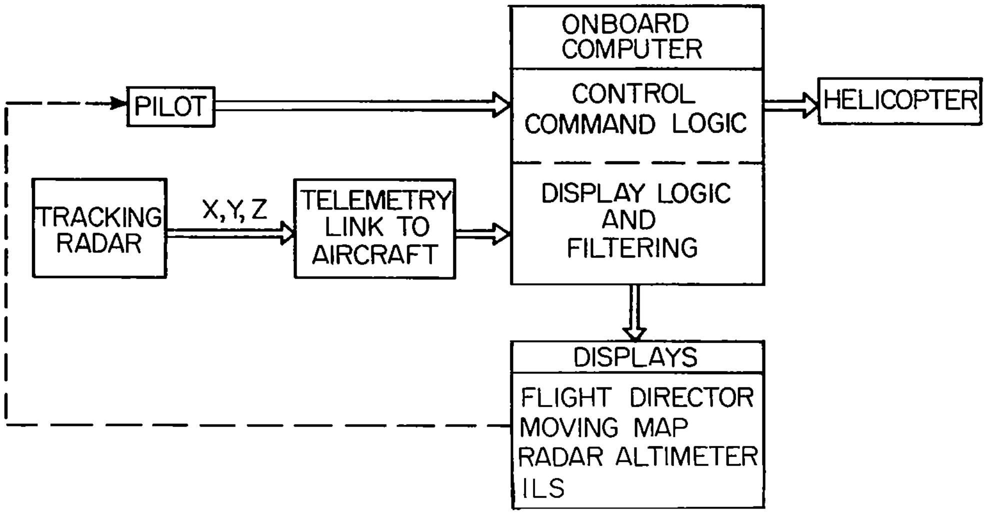

Figure 1.- Signal flow diagram.

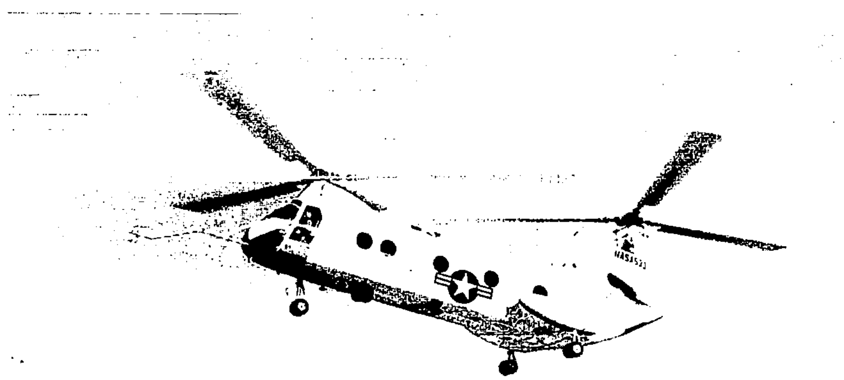

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Figure 2.- Research helicopter.

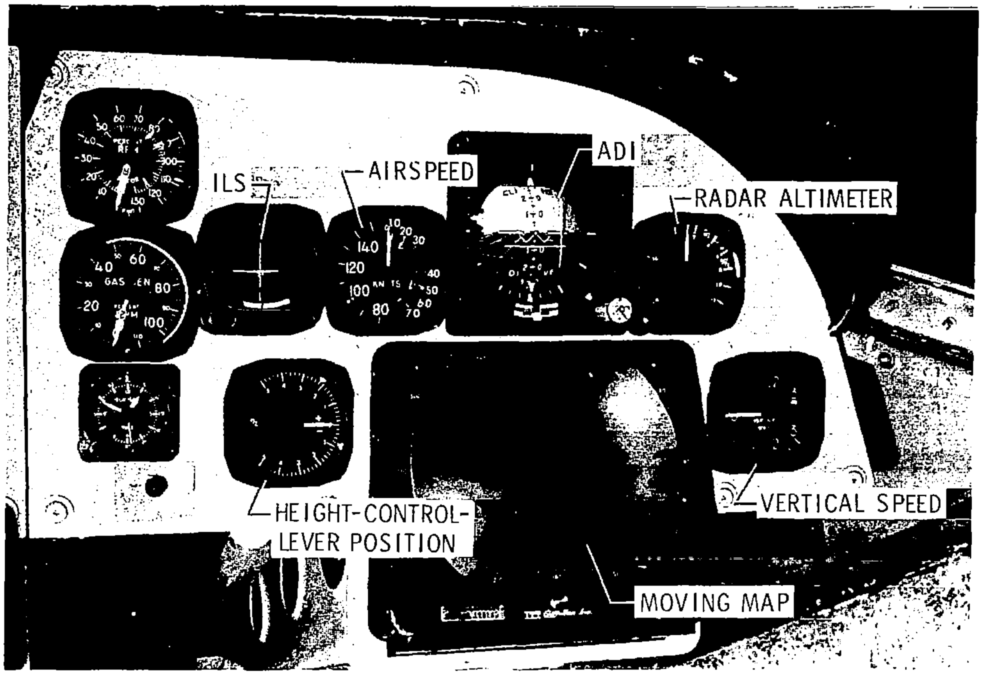

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Figure 3.- Instrument panel.

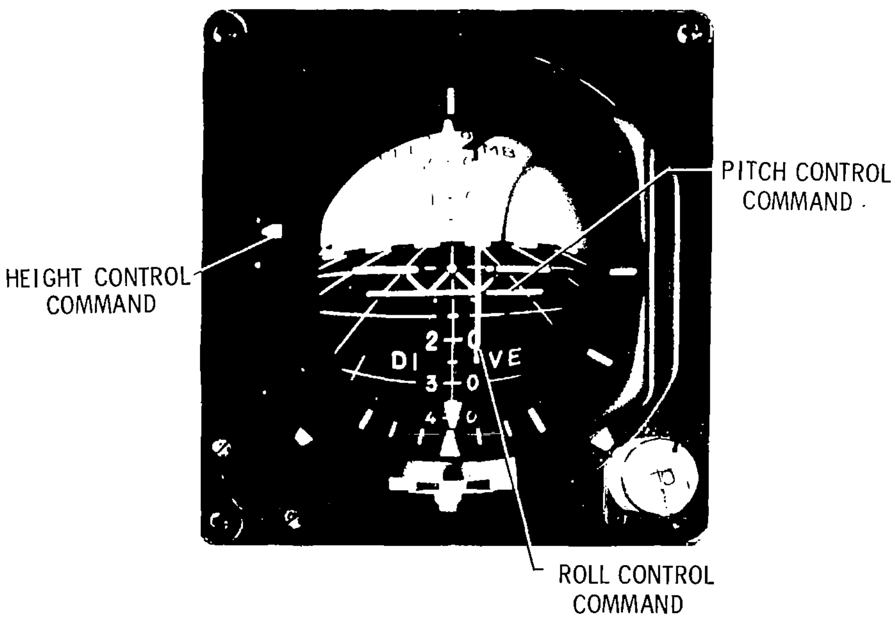

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Figure 4.- Attitude-director indicator.

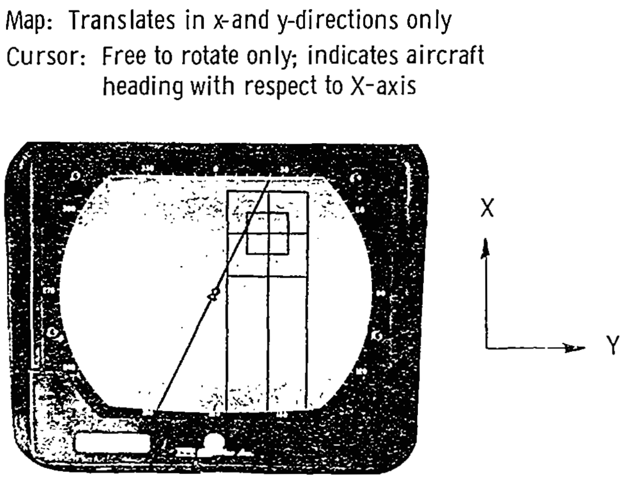

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Figure 5.- Moving-map display.

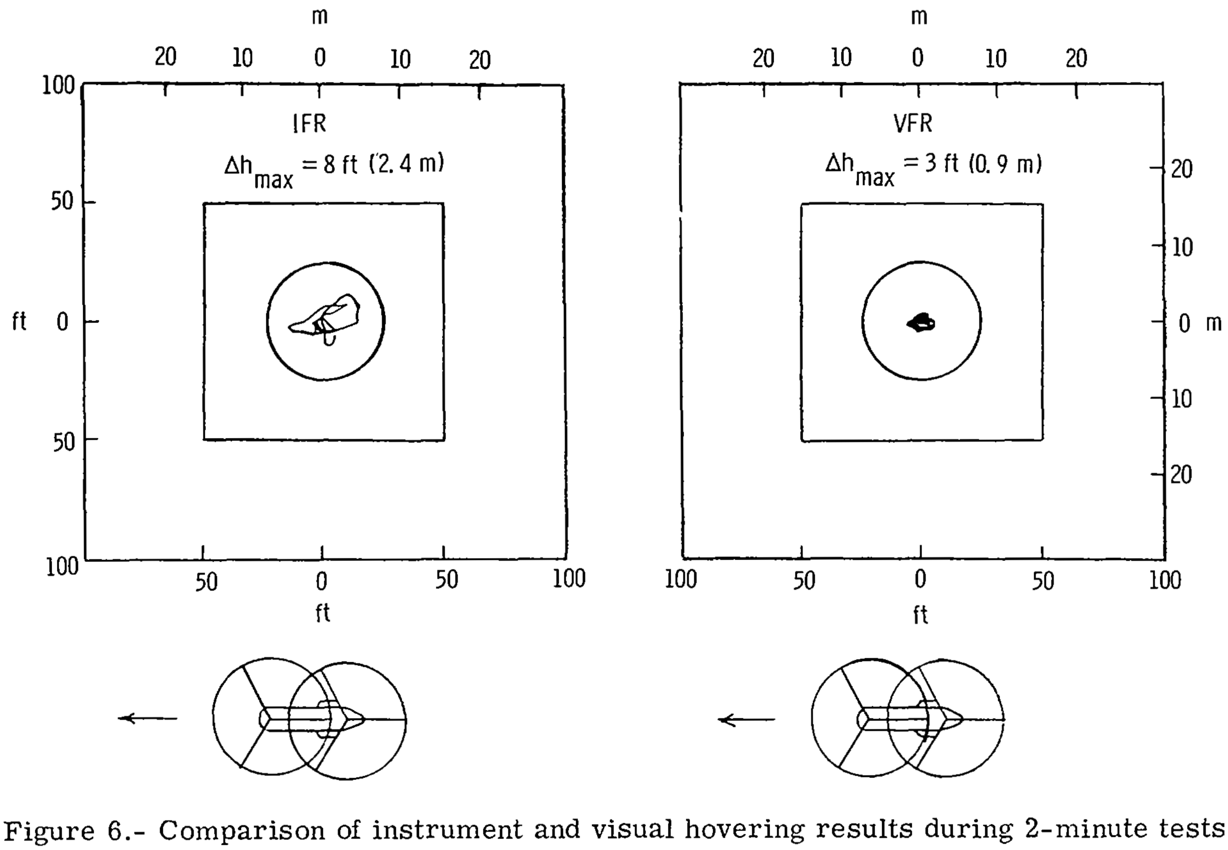
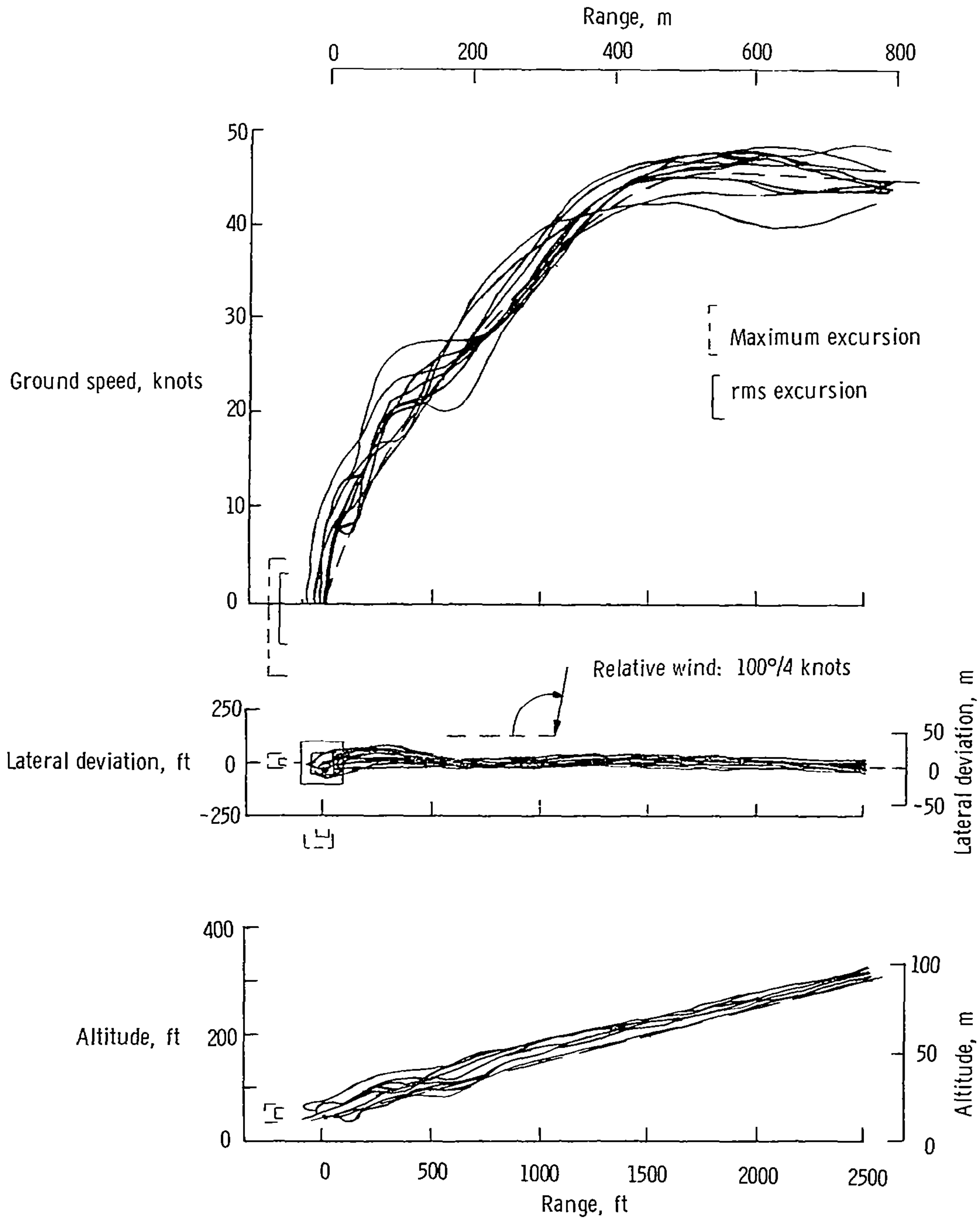
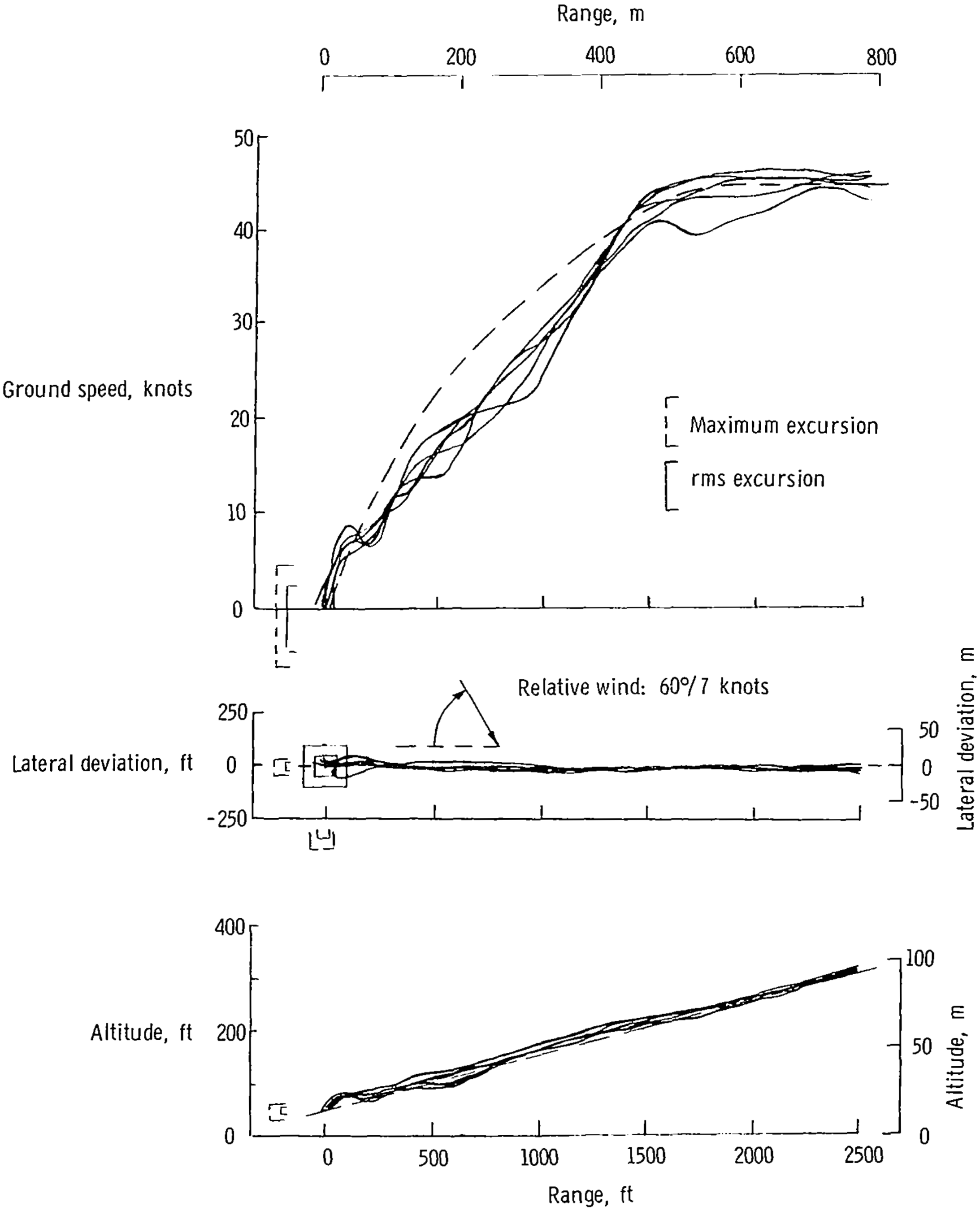


Figure 6.- Comparison of instrument and visual hovering results during 2-minute tests.



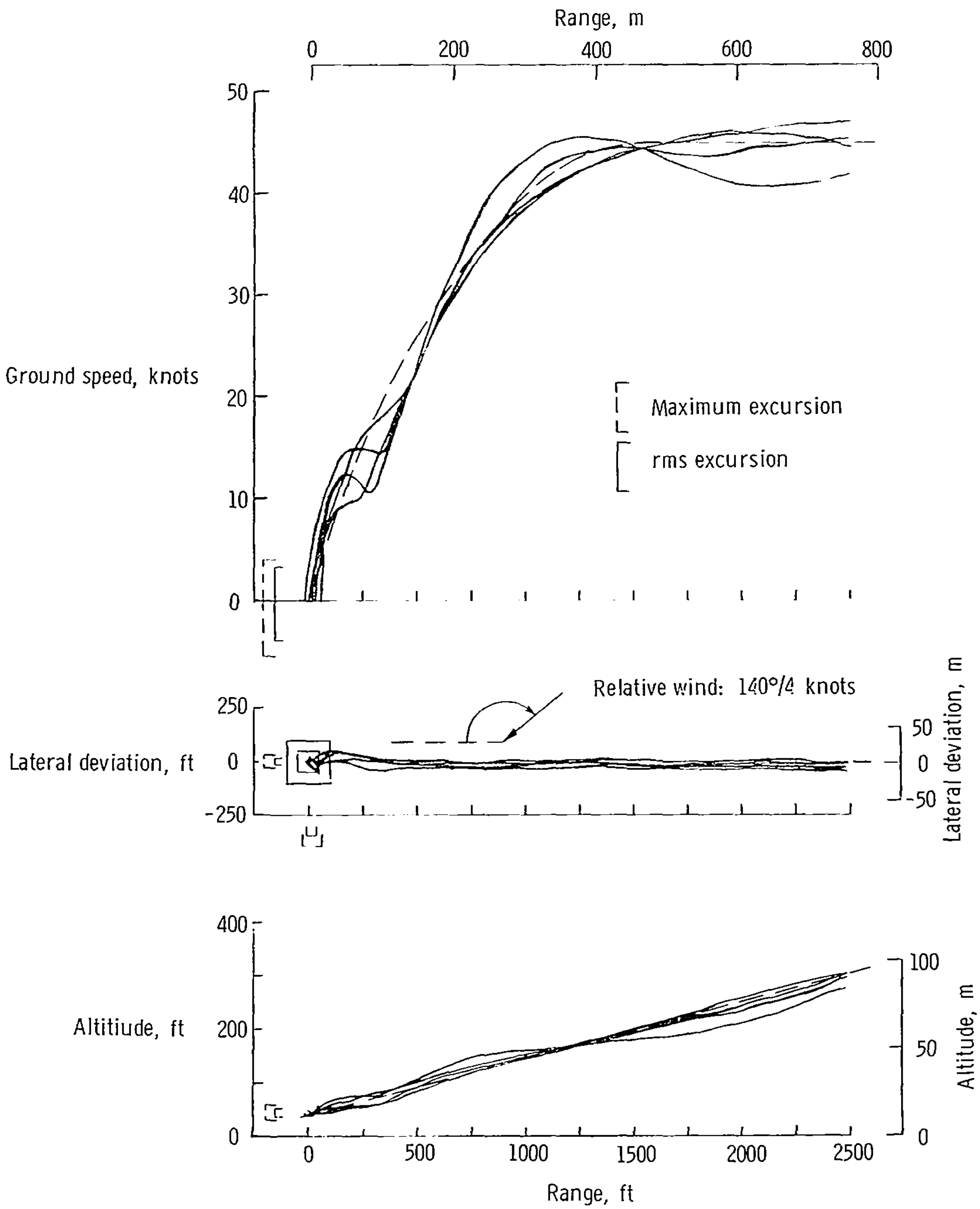
(a) Pilot A; 0.06g deceleration.

Figure 7.- Decelerating approach performance using most refined system.



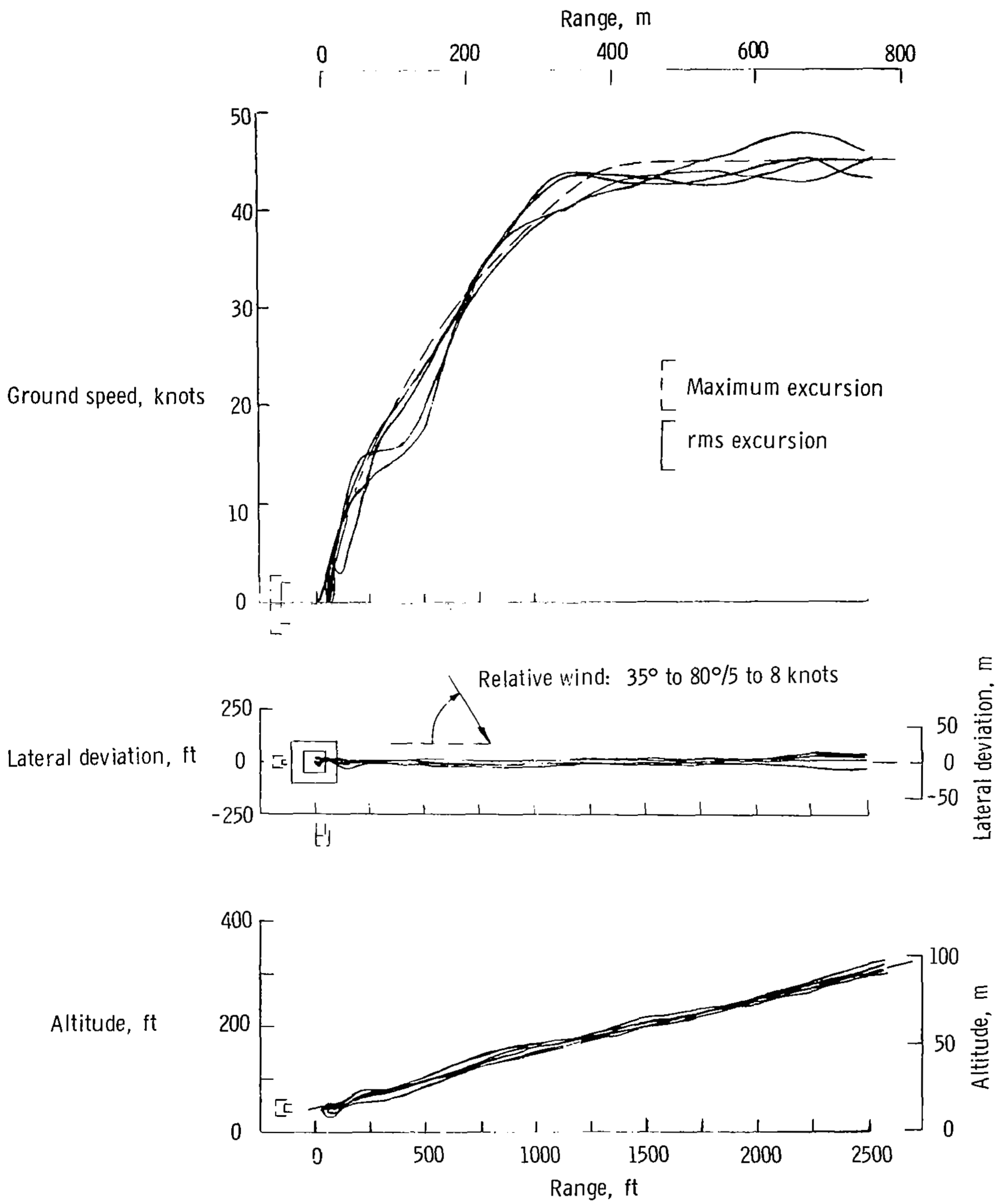
(b) Pilot B; 0.06g deceleration.

Figure 7.- Continued.



(c) Pilot A; 0.08g deceleration.

Figure 7.- Continued.



(d) Pilot B; 0.08g deceleration.

Figure 7.- Concluded.

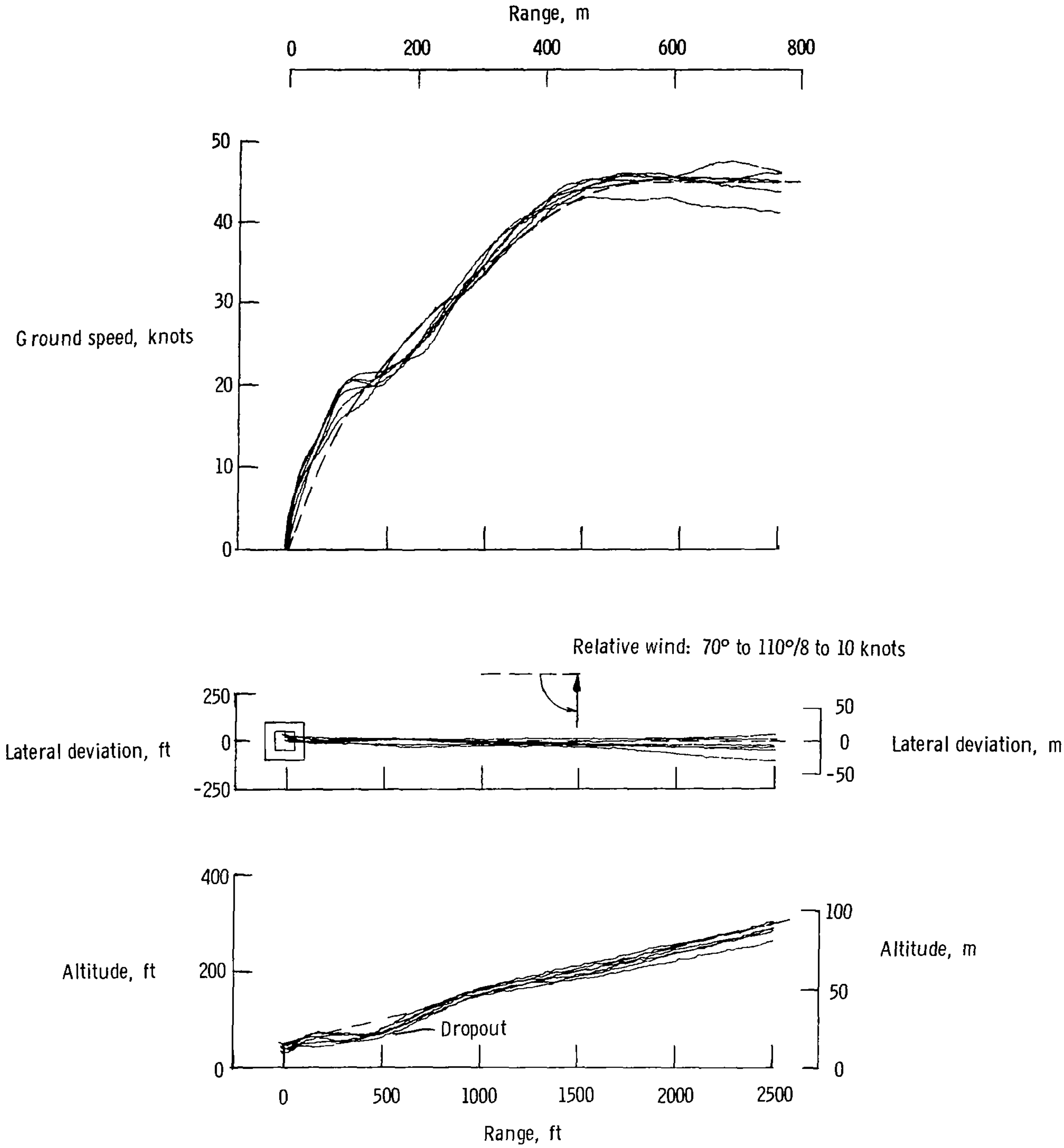


Figure 8.- Illustration of glide-path dropout experienced during early development of the flight-director logic. Pilot A; 0.06g deceleration.

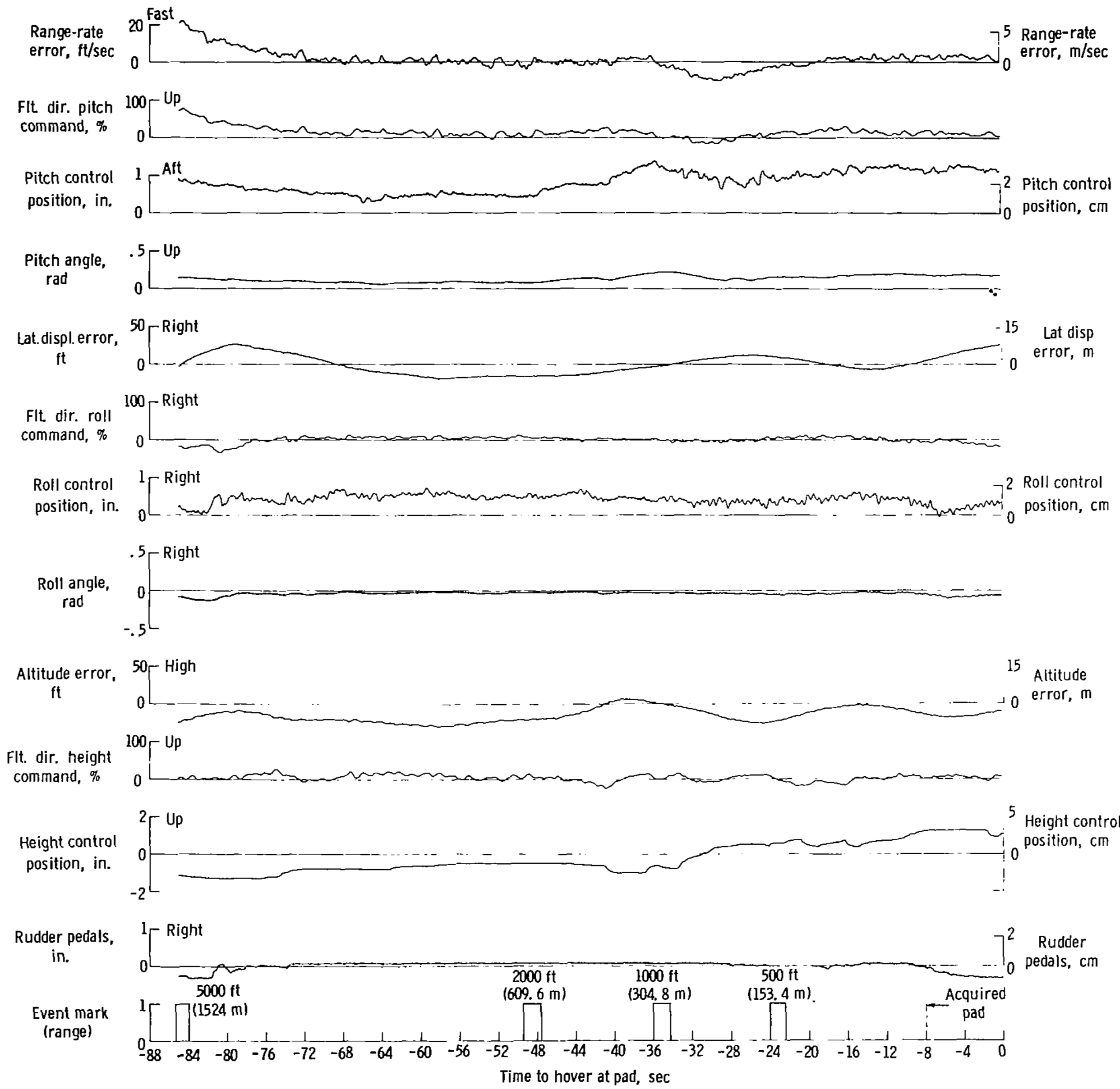


Figure 9.- Time history of typical decelerating approach (0.06g).

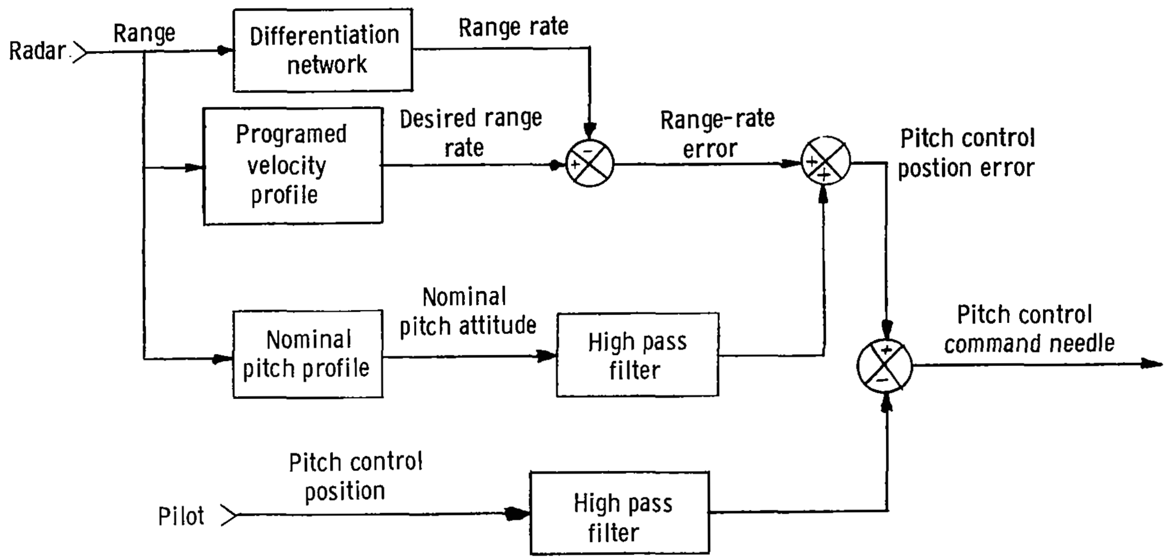


Figure 10.- Pitch-command logic.

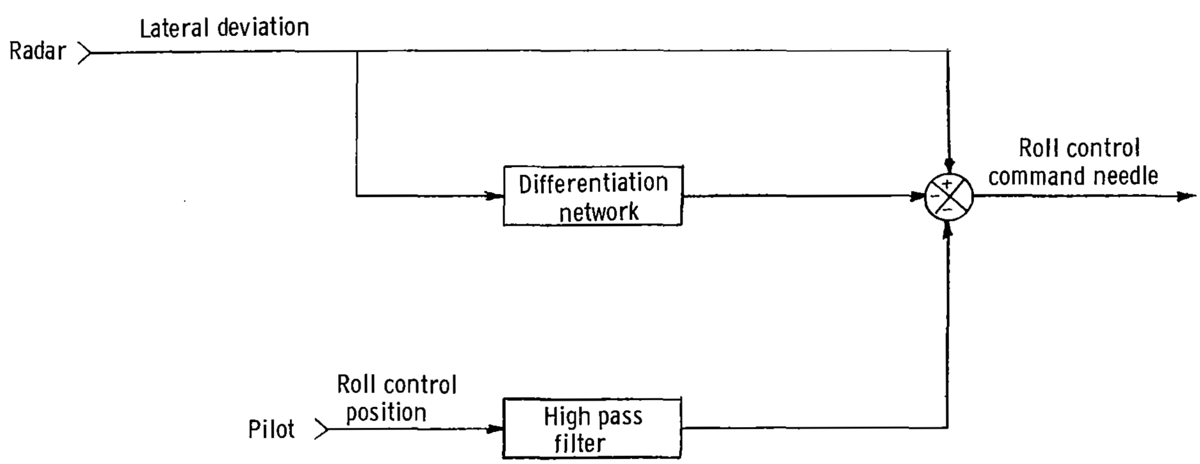


Figure 11.- Roll-command logic.

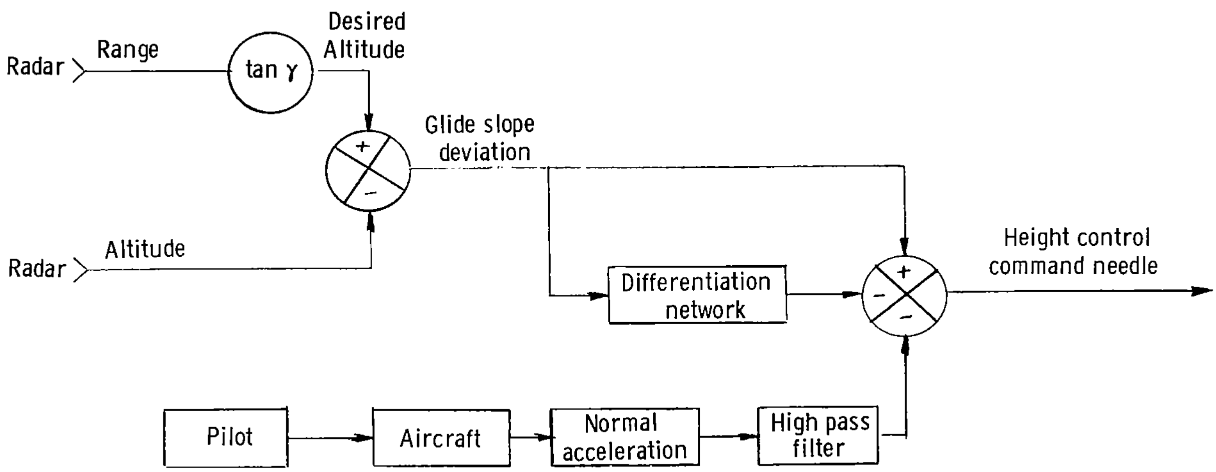
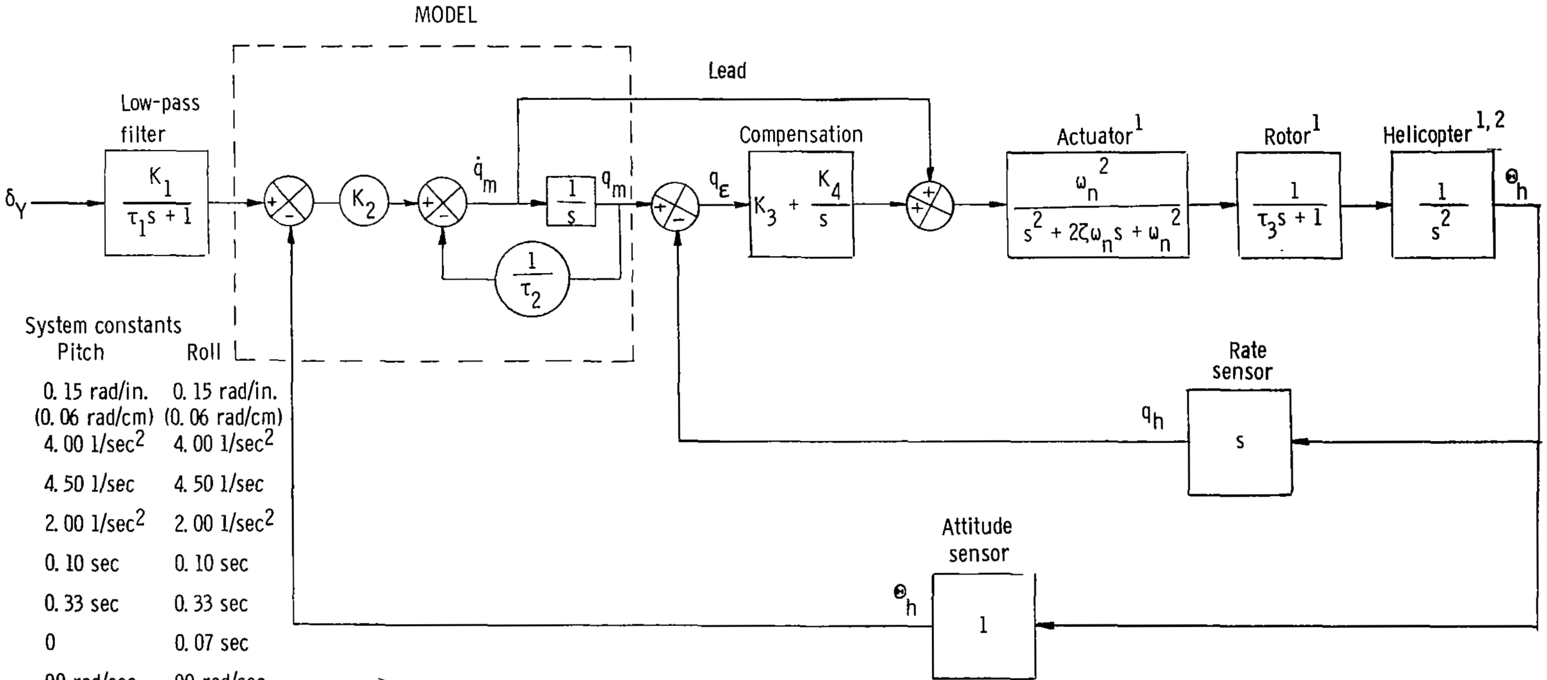


Figure 12.- Height-control-command logic.



System constants

	Pitch	Roll
K_1	0.15 rad/in. (0.06 rad/cm)	0.15 rad/in. (0.06 rad/cm)
K_2	4.00 $1/sec^2$	4.00 $1/sec^2$
K_3	4.50 1/sec	4.50 1/sec
K_4	2.00 $1/sec^2$	2.00 $1/sec^2$
τ_1	0.10 sec	0.10 sec
τ_2	0.33 sec	0.33 sec
τ_3	0	0.07 sec
ω_n	90 rad/sec	90 rad/sec
ζ	0.75	0.75

[1] Gain of this element included in gain of feedback compensation.

[2] Inherent stability characteristics of the research helicopter have been cancelled by feedback loops that are not shown.

Figure 13.- Block diagram of control system. Diagram for pitch only; gains shown for pitch and roll.

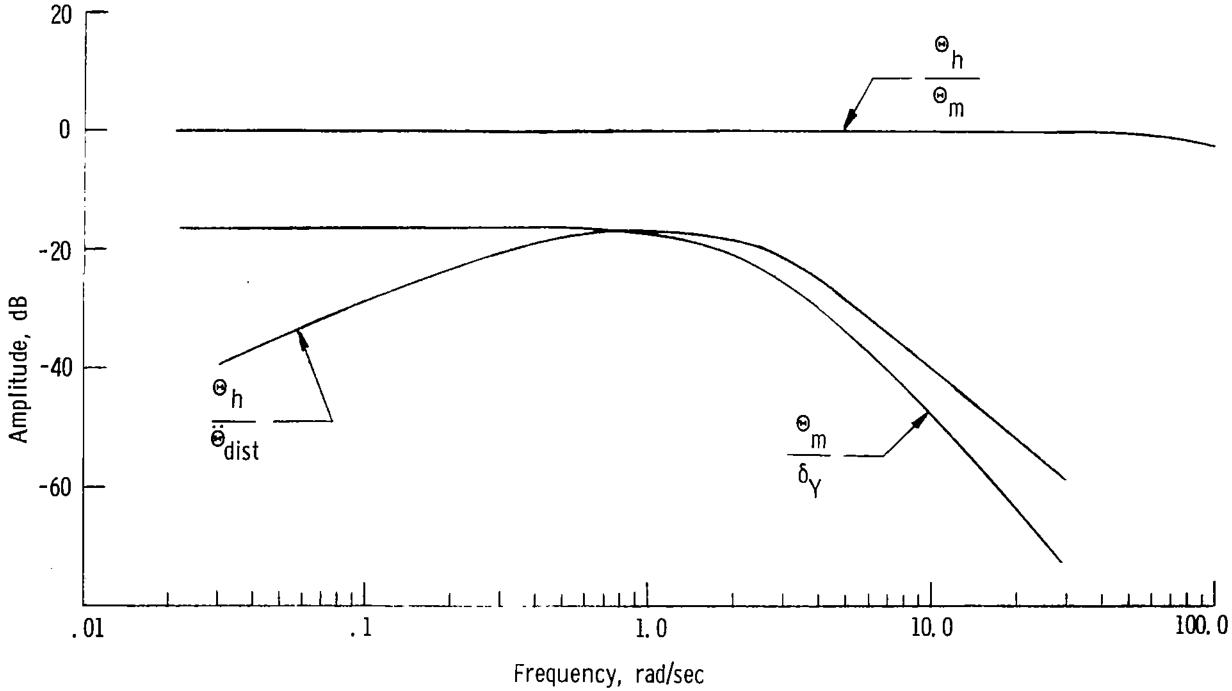


Figure 14.- Frequency response for pitch control loop.

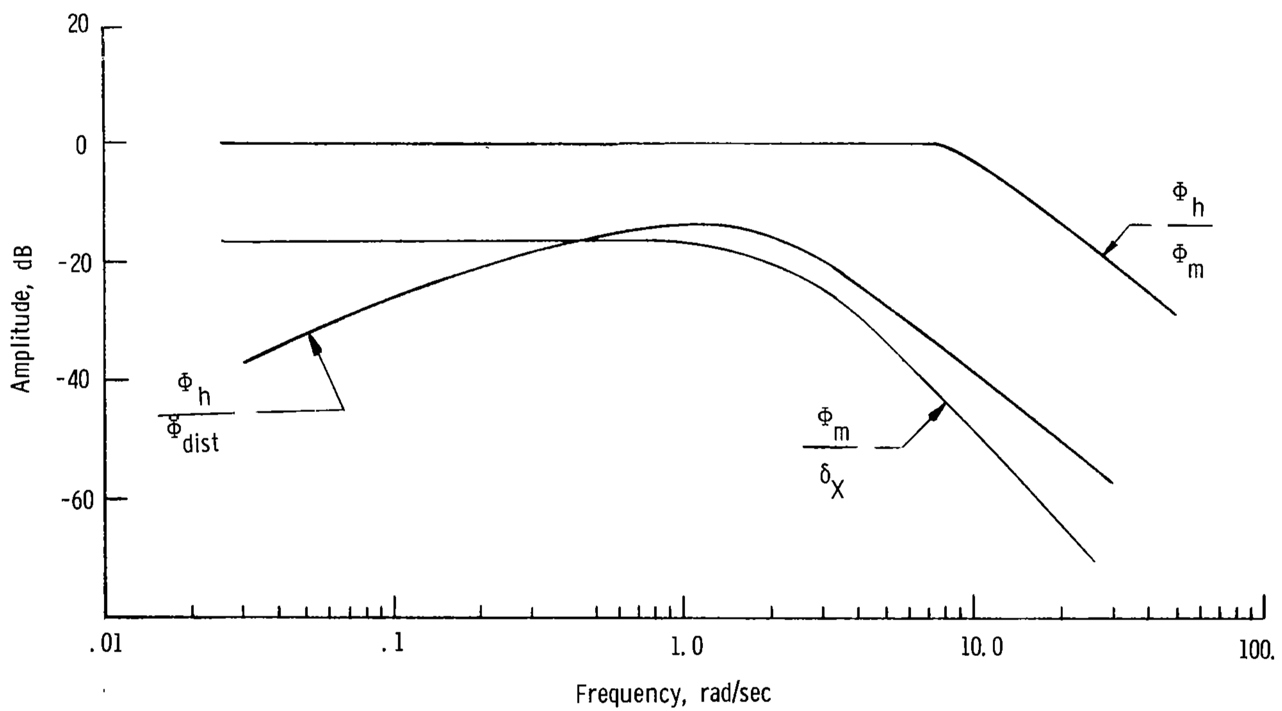


Figure 15.- Frequency response for roll control loop.

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